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RESEARCH MEMORANDUM

SOME OBSERVATIONS OF FLOW AT THE THROAT OF A TWO-DIMENSIONAL DIFFUSER AT A MACH NUMBER OF 3.85

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SOME OBSERVATIONS OF FLOW AT THE THROAT OF A TWO-DIMENSIONAL DIFFUSER AT A MACH NUMBER OF 3.85

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SUMMARY

An experimental investigation was conducted at a Mach number of 3.85 in the Lewis 2- by 2-foot supersonic wind tunnel to study the flow patterns at the throat of a two-dimensional single-shock diffuser and to evaluate qualitatively several schemes for improving the turning conditions. Schlieren observations were made for supercritical inlet operation and for conditions of maximum total-pressure recovery. The angle of attack of the model was limited to zero.

With a near maximum turning at the cowl lip, a large local flow separation, caused by shock-boundary-layer interaction, occurred immediately downstream of the turn on the opposite surface during supercritical inlet operation. This separation was modified to a large degree by the local application of wall suction and was virtually eliminated by a relocation of the impinging shock from the cowl lip at a point immediately downstream of the turn. The use of a ram-type boundary-layer scoop just ahead of the turn or of a shock-cancellation surface downstream of the turn failed to improve the separation condition. With the back pressure adjusted for maximum total-pressure recovery, the terminal shock was observed to be made up of a complex system of shock waves instead of a single "normal" shock.

INTRODUCTION

In order to effect the design of a low-drag inlet configuration, it is often desirable to turn the flow rapidly back in the axial direction to achieve a minimum projected frontal area on the cowl. The problem of turning the flow is generally complicated by boundary-layer considerations and shock-boundary-layer interactions, which can, if not treated properly, result in separation and otherwise poor entry conditions to the subsonic portion of the diffuser. Thus, in the design of an inlet, any gains in the form of a reduced drag, derived from a large rate of turning, must be weighed against any concomitant losses in the efficiency of the diffusion process resulting from a poor entry of the flow at the throat.

Accordingly, the present investigation was undertaken at the NACA Lewis laboratory in an effort to acquire further insight of the turning problem. Schlieren observations were made of the flow patterns at the throat of a two-dimensional single-shock diffuser in order to evaluate qualitatively the effects of several methods for improving the flow conditions. The following design variations were studied: (1) the use of a shock-cancellation surface, (2) the application of local suction after the turn, (3) the installation of a ram-type boundary-layer scoop ahead of the turn, and (4) a relocation of the impinging shock generated by the cowl lip.

APPARATUS AND PROCEDURE

The experimental investigation was performed in the Lewis 2- by 2-foot supersonic wind tunnel at a Mach number of 3.85 and at a simulated pressure altitude of 108,000 feet. The tunnel air was maintained at a temperature of $200^{\circ} \pm 5^{\circ}$ F and at a dew-point temperature of $-15^{\circ} \pm 10^{\circ}$ F. Based on the maximum inlet capture depth (2.56 in.), the test Reynolds number was 220,000.

As illustrated schematically in figure 1(a), the model had a 10-inch span, a 4-inch maximum depth, and a chord of 46.16 inches. An adjustable exit plug, mounted at the rear of a simulated combustion chamber, was used to vary the diffuser back pressure. Glass sideplates were installed at the sides of the compression wedge to permit schlieren observations of the flow patterns and to maintain the two dimensionality of the flow into the inlet. Pressure instrumentation (fig. 1(b)) consisted of pitot and static tubes mounted on rakes just upstream of the variable exit. The pressure rake at the entrance, which may be observed in some of the subsequent schlieren photographs, was not used in the interpretation of the data.

The basic inlet configuration consisted of a 25° wedge, positioned so that the oblique shock would just intercept the cowl lip and involved external supersonic compression only (no internal contraction). An arbitrary turning radius of 0.75 inch was used on the lower turning surface. In order to obtain a near maximum turning of the flow at the cowl lip (within 3° of the detachment angle), the upper surface of the subsonic diffuser was inclined 3° above the horizontal. In order to vary the rate of subsonic diffusion, the angular position of the lower surface downstream of the turn could be set at either 3° or 9° with the horizontal.

To this basic design several modifications (fig. 1(c)) were made. The first modification incorporated the use of shock-cancellation surfaces with the expansion angle set equal to once and twice the strength of the compression wave emanating from the cowl, 22° and 44°, respectively. The second modification involved the application of local suction by venting the cavity below the compression surface to free-stream static pressure and then installing two rows of 1/8-inch-diameter staggered holes with approximately 3/16 inch between spanwise centers
2645 and located immediately downstream of the turn. For the third modification, a ram-type boundary-layer scoop was formed by depressing the initial wedge surface $1\frac{1}{2}^\circ$ and placing a sharp leading edge on the upper surface of the scoop which was located immediately upstream of the turn. The capture height of the scoop was approximately 0.1 inch above the upstream compression surface. Finally, the compression shock originating at the cowl lip was relocated to impinge on the lower surface immediately downstream of the turn. This was accomplished by moving the cowl lip down along a line corresponding to the theoretical leading-edge shock wave. In doing this, an internal contraction ratio of 1.13 resulted (maximum allowable contraction ratio, 1.245).

Schlieren photographs and pressure data were recorded over the range of exit areas for an angle of attack of zero.

RESULTS AND DISCUSSION

Schematic representations and schlieren photographs of the flow patterns near the diffuser throat with supercritical operation are shown in figure 2. For clarity, solid lines were used to represent compression waves; dashed lines, expansion waves; and curled lines, regions of flow separation.

As an initial reference condition, observations were made of the flow turning without the influence of the cowl and are presented in figure 2(a). As would be expected, the flow made the turn with no evidence of any separation.

With the cowl installed and the lower surface adjusted to yield a 12° divergence angle in the subsonic portion of the diffuser, schlieren photographs were taken during supercritical engine operation and the resultant flow pattern is illustrated in figure 2(b). A large local flow separation occurred immediately downstream of the turn and was caused by a high pressure from the impinging shock (originating at the cowl lip) feeding back through the boundary layer. As ordinarily experienced in oblique-shock-boundary-layer interactions, reattachment of the flow occurred after the point of interaction between the compression wave from the cowl and the boundary of the separated region.

With this configuration a maximum total-pressure recovery of 0.17 was obtained (theoretical recovery, based solely on calculated shock losses, 0.34). Corresponding flow patterns obtained under maximum-pressure-recovery conditions will be illustrated and discussed later.

The lower surface of the subsonic diffuser was then adjusted for a 6° divergence angle. In general, the flow pattern (fig. 2(c)) was quite similar to that obtained with the 12° divergence angle; however, the area of the separated region, as viewed by the schlieren apparatus, appeared to be somewhat smaller. One indication of the separation was 2645
given by the fact that the included angle of the expansion fan at the turn was less than that required theoretically and observed experimentally (fig. 2(a)) for the complete expansion of the flow around the corner. With the separation extending forward to the throat, the turning angle was effectively reduced. With the change in subsonic diffuser angle from 12° to 6°, the maximum total-pressure recovery was improved to 0.21.

Another inlet configuration included the use of a shock-cancellation surface, the purpose of which was to set up a flow expansion of sufficient strength to cancel the impinging compression shock emanating from the cowl lip. Schlieren observations indicated no improvement at all. Apparently, the flow was initially separated during the starting process by the diffuser "normal" shock. As this "normal" moved downstream, the reflected shock from the cowl intersected the separation zone and supplied the necessary pressure-rise to sustain it. Actually, there exists some question as to whether or not this device would be effective in reducing the separation difficulty even with an initially attached flow at the throat.

An attempt to reduce the local flow separation after the turn was made with the application of suction immediately downstream of the turn. As illustrated in figure 2(e), the cross-sectional area of the separated flow was markedly reduced with wall suction. This was illustrated by the large increase in the included angle of the expansion fan at the turn compared with that previously observed for the case without suction. As qualitatively illustrated herein and used in reference 1, the method of applying suction locally can be effectively used to modify or control flow separation. Associated with this improvement in the supercritical-flow condition near the diffuser throat, an increase in the maximum total-pressure recovery to 0.23 was realized.

In order to observe the effect of boundary-layer removal at the end of the compression surface, a ram-type scoop was installed just upstream of the turn. As illustrated in figure 2(f), removing the boundary layer just ahead of the favorable pressure gradient on the turn did not avoid the separation difficulty downstream of the turn. The resulting separation pattern and the value of maximum total-pressure recovery were the same as that obtained without a scoop. As shown in the schlieren photographs, the use of leading-edge roughness did not appear to have any effect either on the separation pattern or on the maximum recovery value. However, it was observed that with a smooth leading edge the boundary layer seemed to thicken or separate just ahead of the scoop and that with a rough leading edge the boundary layer seemed to thin or neck down just ahead of the scoop.

Another design variation included a modified cowl, one designed so
2645 that the reflected shock from the lip would impinge on the lower surface at a point immediately downstream of the favorable pressure gradient on the turn. As shown in the schlieren photographs of figure 2(g), the local flow separation, previously described, was practically eliminated. With this configuration a maximum total-pressure recovery of 0.26 was obtained; however, the modified cowl created a slight internal contraction and, consequently, the corresponding theoretical value of maximum recovery was increased to 0.38. Again there was little or no effect of leading-edge roughness on the value of maximum total-pressure recovery. With a smooth leading edge, there appeared to be some thickening or a slight separation of the laminar boundary at and just ahead of the impinging shock; whereas, with a rough leading edge, the boundary layer appeared thicker over the entire surface of the wedge but showed no indication of any flow separation in negotiating the turn.

Schlieren photographs of the inlet flow patterns during operation at maximum total-pressure recovery are presented in figure 3. In general, there was a rather poor definition of the shock system at or downstream of the throat. In each case, a slight oscillation of the flow pattern at the cowl lip was encountered. It was also observed that in no case could a single normal shock pattern be formed at or near the diffuser throat; the terminal shock consisted, rather, of a system of shock waves. The configurations with a clean or smooth leading edge (figs. 3(a) to 3(c)) indicated a thickening or separation of the boundary layer along the compression surface just upstream of the turn; this did not appear to be true of the case where roughness was applied (fig. 3(d)).

As would be expected on the basis of the criterion given in reference 2, these inlet configurations (all of which had the leading-edge shock located at the cowl lip) indicated no stable range of subcritical operation. In every case, the "buzz" pattern appeared quite similar to that obtained with typical axially symmetric nose inlets.

SUMMARY OF RESULTS

Experimental observations of the flow patterns in the vicinity of the throat of a two-dimensional single-shock diffuser yielded the following qualitative results at a Mach number of 3.85:

1. Local flow separation, caused by shock-boundary-layer interaction and located immediately downstream of the expansion-turn, was controlled to some degree by the application of wall suction.

2. With the oblique shock from the cowl surface located at a point immediately downstream of the turn, local flow separation was virtually eliminated.

3. The use of either a ram-type boundary-layer scoop just ahead of the turn or a shock-cancellation surface downstream of the turn failed to improve the local separation condition. 2645

4. In no case could a single normal-shock pattern be formed at or near the throat; instead, the terminal shock consisted of a complex system of shock waves.

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1. Connors, James F., and Woollett, Richard R.: Performance Characteristics of Several Types of Axially Symmetric Nose Inlets at Mach Number 3.85. NACA RM E52I15, 1952.

2. Ferri, Antonio, and Nucci, Louis M.: The Origin of Aerodynamic Instability of Supersonic Inlets at Subcritical Conditions. NACA RM L50K30, 1951.

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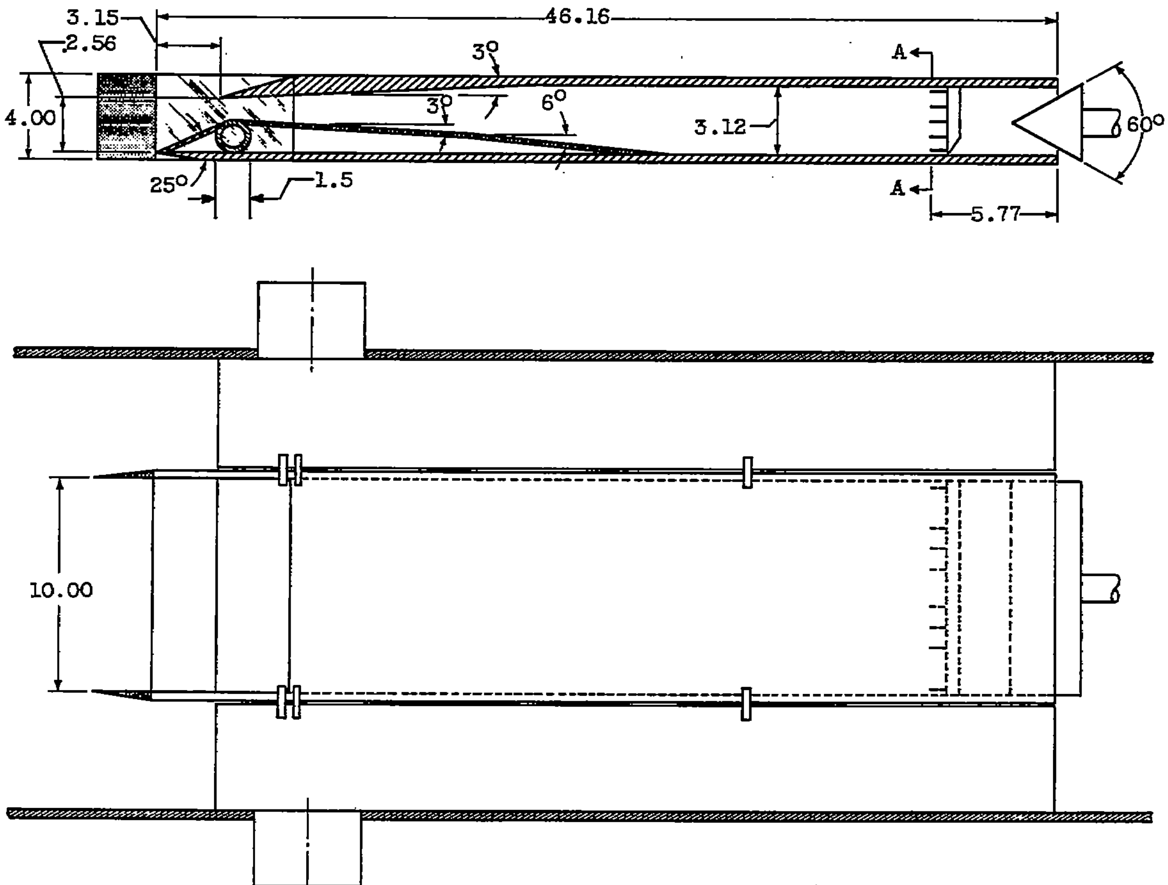
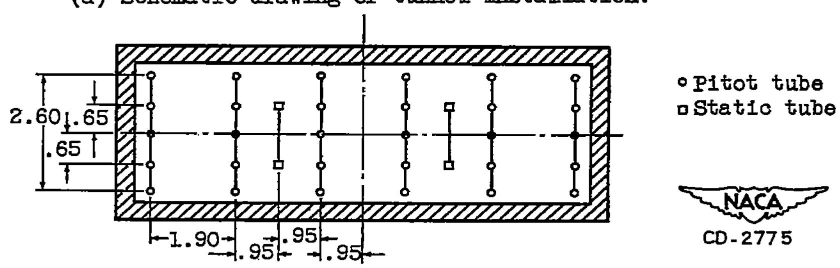


(a) Schematic drawing of tunnel installation.



(b) Instrumentation (pressure rake, section A-A).

Figure 1. - Experimental model.

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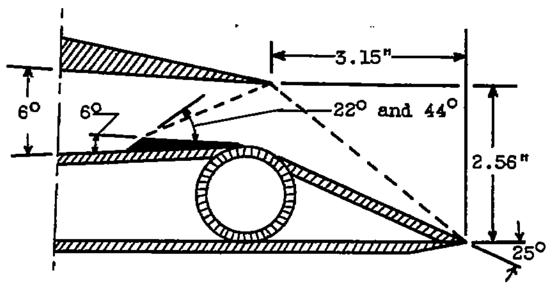
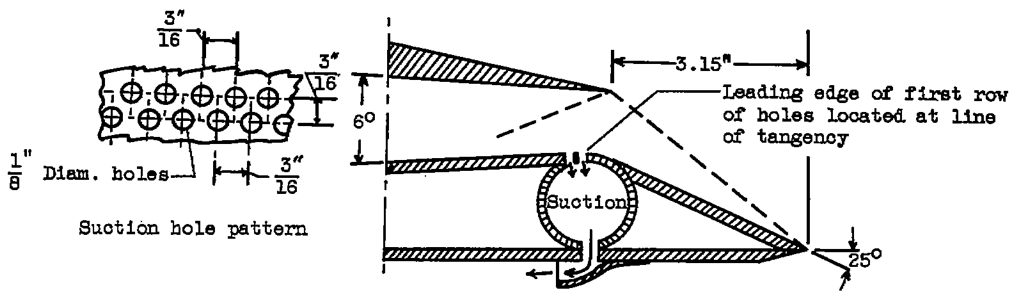
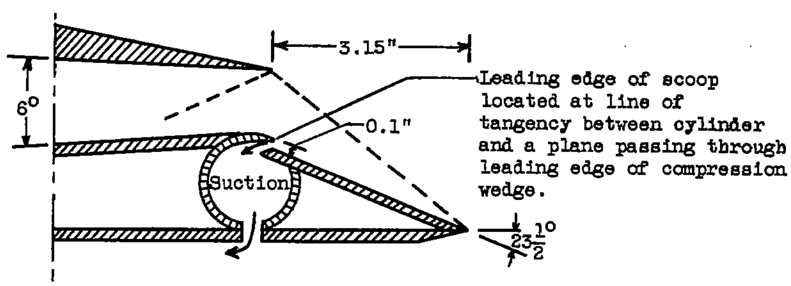
Shock-cancellation surfaces



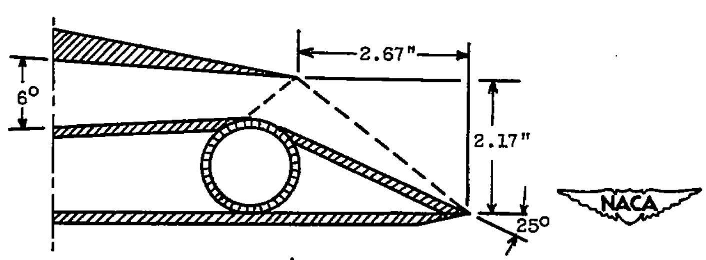
Local suction after turn



Ram-type scoop ahead of turn

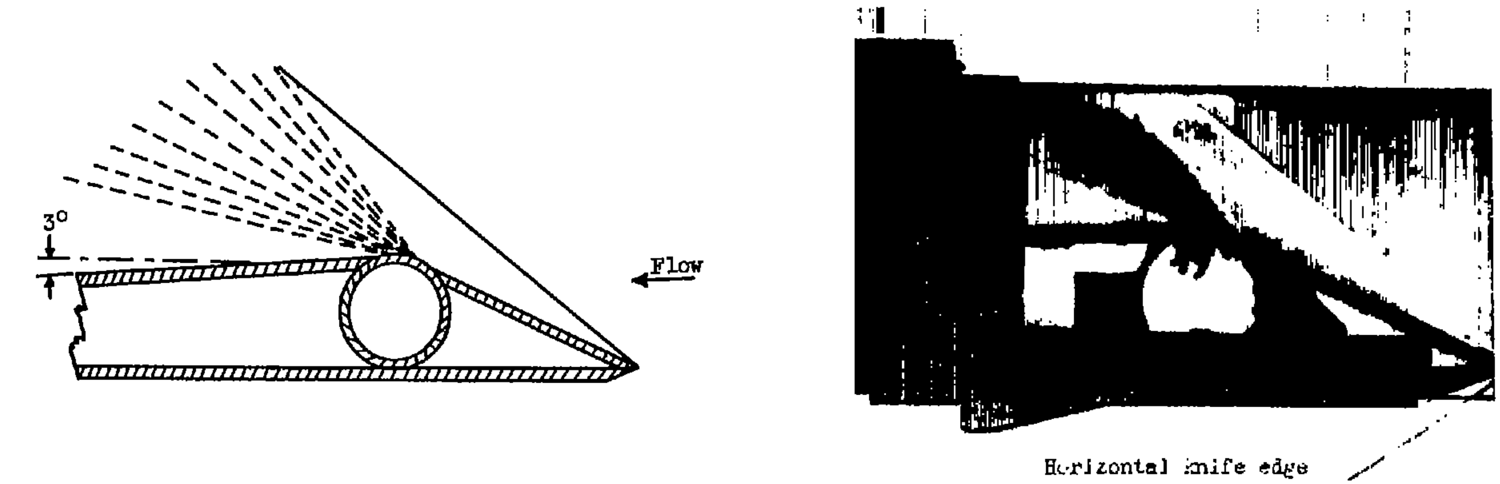


Cowl modified for shock relocation

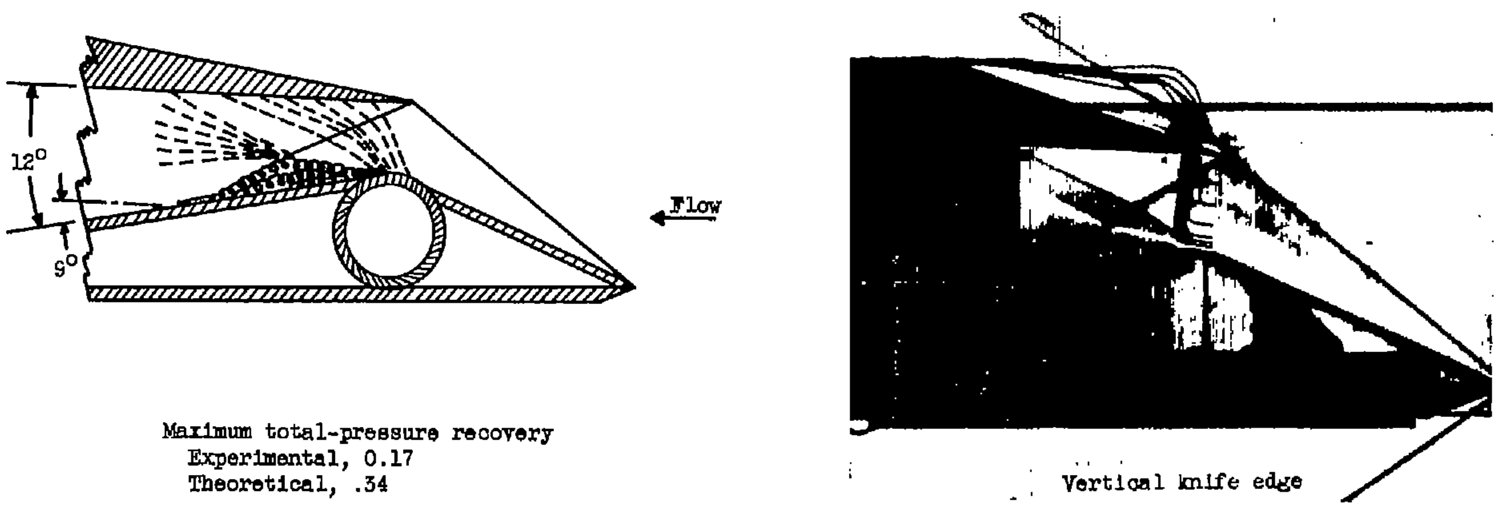


(c) Inlet modifications.

Figure 1. - Experimental model.



(a) Without influence of cowl.



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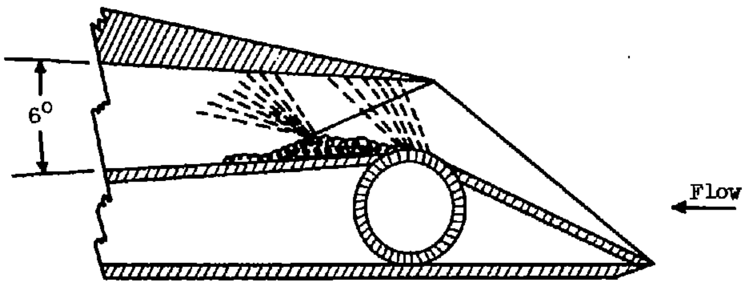
(b) Subsonic diffuser angle, 12°.

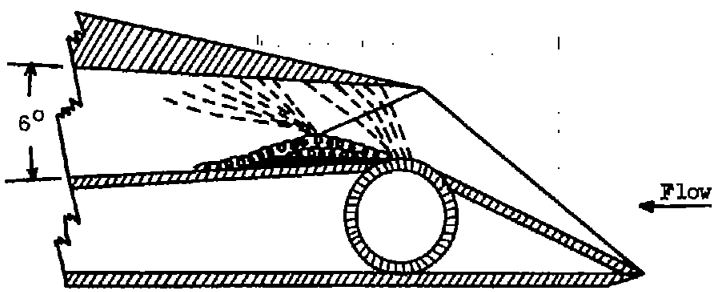
Figure 2. - Schematic representation and schlieren photographs of flow near diffuser throat with and without cowl and with supercritical operation.

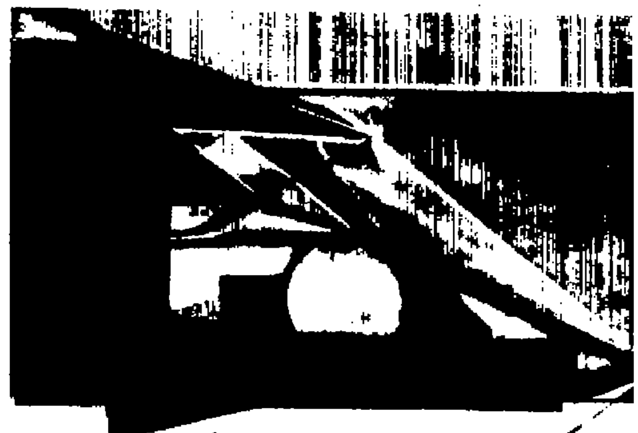


Maximum total-pressure recovery
Experimental, 0.21
Theoretical, .34

Horizontal knife edge

(c) Subsonic diffuser angle, 6°.





Horizontal knife edge

(d) With shock-cancellation surface.

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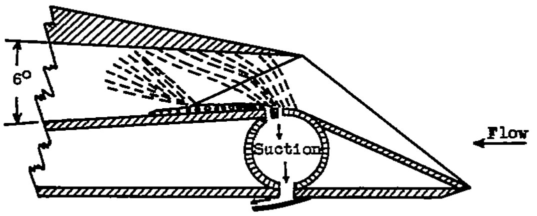
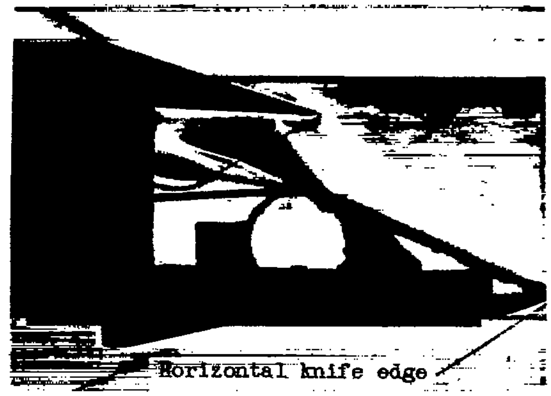
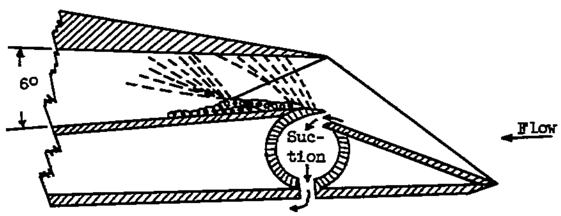
Figure 2. - Continued. - Schematic representation and schlieren photographs of flow near diffuser throat with and without cowl and with supercritical operation.



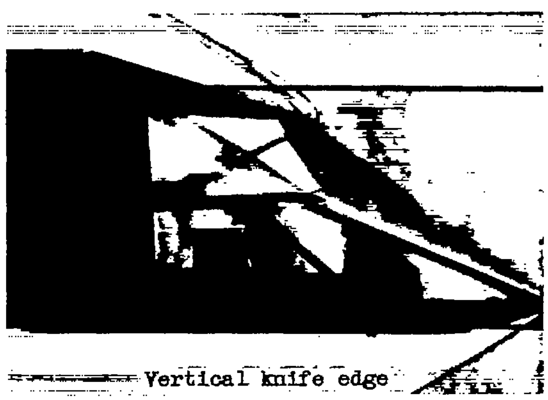
Maximum total-pressure recovery
Experimental, 0.23
Theoretical, .34



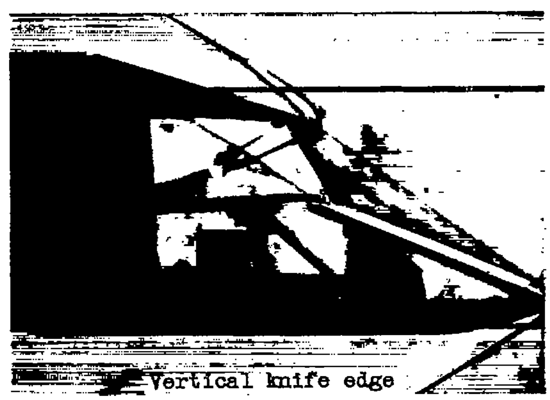
(e) Application of suction after turn.



Maximum total-pressure recovery
Experimental, 0.21
Theoretical, .34



Smooth leading edge



Rough leading edge

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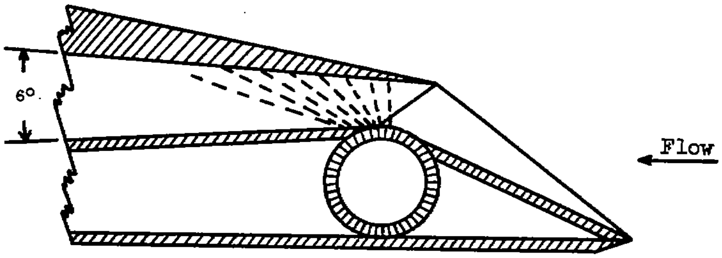
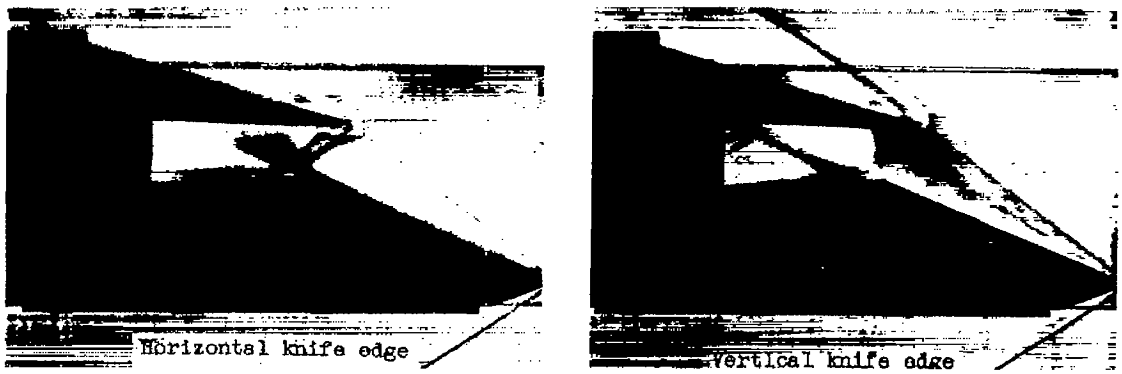
(f) Ram-type boundary-layer scoop ahead of turn.

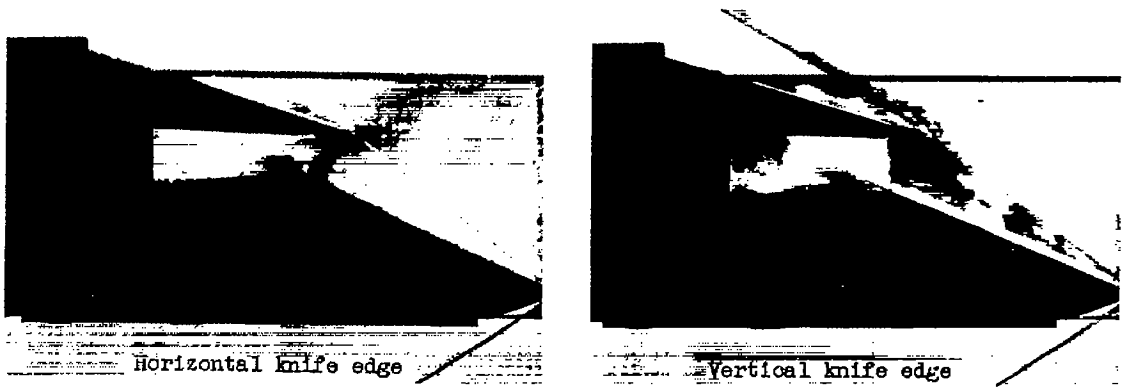
Figure 2. - Continued. Schematic representation and schlieren photographs of flow near diffuser throat with and without cowl and with supercritical operation.



Maximum total-pressure recovery
Experimental, 0.26
Theoretical, .38



With smooth leading edge



With rough leading edge

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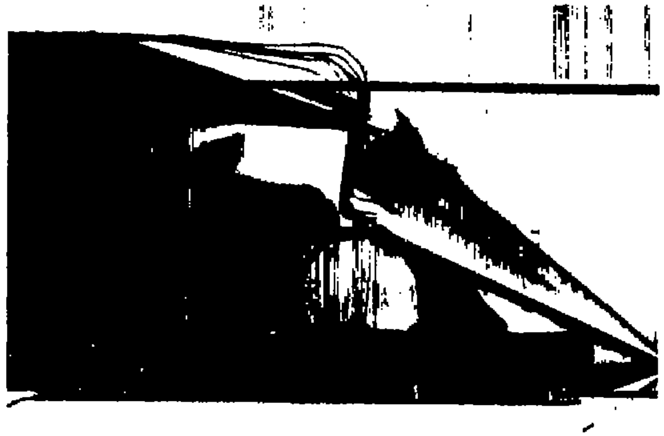
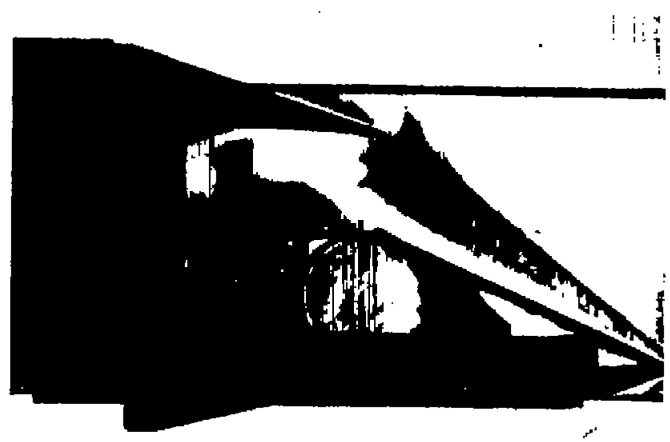
(g) Relocation of reflected shock from cowl.

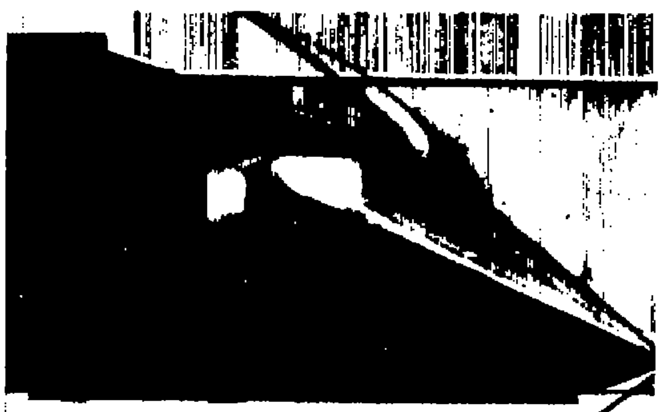
Figure 2. - Concluded. Schematic representation and schlieren photographs of flow near diffuser throat with and without cowl and with supercritical operation.



(a) Subsonic diffuser angle, 6°.



(b) Application of suction immediately downstream of turn.



(c) Relocation of reflected shock and smooth leading edge on wedge.

(d) Relocation of reflected shock and rough leading edge on wedge.

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Figure 3. - Schlieren photographs of inlet flow patterns during operation at maximum total-pressure recovery.